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A Note on an Acoustic Response During an Engine Nacelle Flight Experiment

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RESPONSE DURING AN ENGINE NACELLE FLIGHT
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ABSTRACT

During a flight test study of the noise effects on laminar flow on the outside surface of a simulated engine nacelle, an intense acoustic response was observed. The aircraft speed at which this signal occurred and the frequency content of the signal fell within the test conditions of the experiment and had to be eliminated prior to continuing. The signal was identified as an aerodynamic excitation of an acoustic mode in the simulated by-pass duct of the nacelle. By modifying the trailing edges of the support struts of the nacelle, the aerodynamic excitation was changed enough to eliminate the resonant response of the offending duct modes, eliminating the unwanted acoustic problem.

SUMMARY

As part of a laminar flow noise experiment, a simulated engine nacelle was mounted on the wing of an OV-1 test aircraft. During the flight test, an uncontrolled high level acoustic signal occurred, interfering with the planned experiment. By modifying the trailing edge of nacelles' internal support struts, the unwanted signal was eliminated allowing the original experiment to be continued without difficulty.

INTRODUCTION

As part of a flight test study into the effects of noise on a laminar flow boundary layer over the exterior surface of an engine nacelle, a flow-through (simulated high-bypass ratio engine nacelle with no engine) nacelle was built and mounted on the wing of an OV-1 test aircraft. Two controllable noise sources, one on the wing outboard of the nacelle and the other inside of the nacelle in a centerbody section, were included in the test configuration. The centerbody of the nacelle was sized to create a duct which simulated the by-pass area of the engine. Support for the centerbody was provided by four struts connecting it to the shell of the nacelle. During the preliminary performance flight test, to check the instrumentation systems and nacelle orientation, all the signals from the internal fluctuating-pressure transducers and the external fluctuating-pressure transducers near the leading edge of the nacelle saturated. The gains of these transducers had been set to measure the maximum levels of the controllable noise sources; however, in flight, these transducers severely overloaded even when the controllable noise sources were turned off. Initial investigations showed the problem to be a tone of 775 Hz and several of its harmonics at very high levels. The tone appeared at aircraft speeds between a nominal 66.9 m/s to 74.8 m/s (130 knots to 145 knots). Both the frequency (particularly the second and third harmonics) and the aircraft speed at which the tone occurred fell in the operating range of the laminar flow experiment and had to be eliminated before testing could advance.

Although there might be several candidate explanations for this phenomena, one likely possibility was the interaction of the vortex wake from the internal support struts with an internal resonant acoustic mode in the inside of the nacelle, such as a

"Parker mode." Using this interpretation as an explanation, a "cure" for the problem was implemented. It is the purpose of this paper to present the data, method of identification of the acoustic mode, and the solution which eliminated the problem.

TEST CONDITIONS

A flow-through nacelle designed to support natural laminar flow on the outside contour was designed, built, and mounted on the wing of an OV-1 aircraft along with an external pod containing a camera and a controllable noise source, figure 1. The nacelle is 210.8 cm (83 inches) long and has a centerbody section, supported by four struts. The centerbody was sized to create a duct simulating the high by-pass duct of a turbofan engine. Details on the geometry of the nacelle are shown in figure 2 and a photograph of the nacelle in figure 3. Inside the centerbody, a second controllable noise source was mounted for use in the noise tests.

Preliminary test flights to check out the aircraft's performance, the instrumentation systems, and the nacelle orientation were conducted by flying the aircraft in straight and level flight at 457.2 meters (1500 feet) and varying the aircraft's speed from a nominal 61.7 m/s (120 knots) to a nominal 77.2 m/s (150 knots). Only the port turboprop engine (on the opposite wing) was used during these check flights to minimize the starboard propeller's slipstream and noise effects on the nacelle. The starboard turboprop engine (on the same wing as the test nacelle) was turned off and the propeller placed in a feathered position. Data from all the transducers were recorded during the speed variation.

INSTRUMENTATION

The aircraft had instrumentation to measure several of the steady-state aircraft performance parameters. In addition, there were 15 Kulite crystal-type fluctuating-pressure transducers (XC5-190) flush-mounted to the surface in the nacelle. Six of them were internal and were rated at ± 103.4 kPa (± 15 psi,) and nine were mounted externally and were rated at ± 34.5 kPa (± 5 psi.) Each of these transducers were calibrated at 130 dB and 1000 Hz, and the output recorded on one of two FM analog tape recorders.

DISCUSSION

The problem.- The gain setting on each of the fluctuating-pressure transducers was set to allow the maximum level signal from either or both of the noise sources mounted on the aircraft to be within the range of the recorder system. These values of the pressure amplitude were determined from ground tests and in a prior configuration had proven to be a very reliable estimate. During the first flight test and even before the noise sources were turned on, all of the interior pressure levels and those recorded near the leading edge of the nacelle on the outside saturated the recorder system. Gain settings were adjusted and on a subsequent flight the signals were

recordable although the levels were far in excess of what was expected. A spectral analysis of the data from one of the internal transducers (typical of all the data), figure 4, shows what occurred during an aircraft speed variation run. Plotted is a stepped, sweep time history of the pressure spectra for the range of aircraft speeds flown. The vertical scale is root-mean-square amplitude in dB (ref. 2×10^{-5} Pascals), the horizontal scale is frequency (0 Hz to 5000 Hz in 12.5 Hz increments), and the slanting scale is increasing time representing, in this case, increasing aircraft speed. As the aircraft speed increases from 61.7 m/s to 66.9 m/s (120 knots to 130 knots), the dominate signals are below 500 Hz and can be attributed to the engine propeller on the port side, while from 500 Hz to 5000 Hz the spectra is relatively flat at a low level. However, at about 66.9 m/s (130 knots,) a sharp rise in the spectra occurs at 775 Hz and several (5) of its harmonics. The peak at 775 Hz and its second harmonic (1550 Hz) remain until the aircraft speed reaches about 74.6 m/s (145 knots) and then completely disappear from the spectra at the higher speeds. Figure 5 shows a single spectra taken from the three-dimensional plot of the data obtained from a microphone mounted inside the nacelle. The overall level is 141 dB, a level higher than that available from the controllable noise sources. Overall levels from the other internal transducers are similar, however, there are not sufficient measurement locations to deduce a mode shape.

Detailed evaluation of the aircraft speed variation tests shows the response to be a high level signal, constant in frequency over a limited aircraft speed range. Unfortunately, both the frequency of this signal and the aircraft velocity at which the phenomena occurred fell into the range of test conditions for the laminar flow test matrix and had to be eliminated before the experiment could continue.

The data suggest that the mechanism was a resonant phenomena (high levels, constant frequency), produced by an aerodynamic exciting function (limited airspeed range of response). One such mechanism that has been identified is a "Parker mode", references 1 and 2. Parker established that a significant acoustic response could be generated by the interaction between the vortex shedding frequency from a blade in a duct and the acoustic properties of the duct. His studies were part of an investigation into turbomachinery noise in which noise might be generated from the flow over the compressor blades or similar bodies (i.e. inlet or outlet guide vanes, support struts for the turbine-compressor) and the by-pass duct of the turbomachine. The flow-through nacelle with its four support struts represents a similar configuration.

ANALYSIS AND TEST

Resonant conditions. - Based on the assumption that a duct resonant mode was excited by wake shedding from the strut, a simple calculation of the duct standing modes in the nacelle was performed. Although the inside of the nacelle is not a simple cylinder (i.e. it is not a constant diameter along the length and there is a centerbody supported by four struts), a simple calculation was made using this assumption. Dimensions of the inside of the nacelle are given in figure 2. The acoustic frequencies, f_{mns} , were calculated using equation 3.24 from reference 3 with an end correction applied to the length of the cylinder to account for the open

ends from reference 4.

$$f_{mns} = f_m \sqrt{1 + \left(\frac{1}{m^2}\right) \left(\frac{L'_x}{R}\right)^2 \left(\frac{\bar{R}_{ns}}{\pi}\right)^2}$$

where

c speed of sound

f_m the m resonant frequency of an open-open cylinder axial mode,
 $f_m = m c / 2 L'_x$

m number of axial half-wavelengths along the cylinder

L_x length of cylinder

L'_x corrected length of cylinder, $L'_x = L_x + 2(.61)R$

R radius of the cylinder

$\frac{\bar{R}_{ns}}{\pi}$ roots of a Bessel equation, Table 1

n number of circumferential waves around the cylinder

s number of radial waves in the cylinder

The inputs and results of these calculations for the first several modes of the cylinder are shown in Table 1, and the results are plotted in figure 6. Figure 6 shows three candidate modes close to the 775 Hz observed in flight. The axial half waves number are $m=1, 2$, and 3 with a circumferential wave number of $n=3$, and a radial wave number of $s=0$.

An experiment was then conducted to determine if an excitation near the trailing edge of the strut could excite one of these three candidate acoustic modes. An acoustic driver was moved along an inboard chord of the strut from 13.97 cm (5-1/2 inches) forward of the trailing edge to 20.3 cm (8 inches) aft of the trailing edge. At three locations in this range, three significant acoustic responses were detected by two microphones inside the nacelle. For a constant input to the driver, the microphones showed peaks at 740 Hz, 775 Hz, and 790 Hz at driver locations 13.97 cm (5-1/2 inches) forward of the trailing edge, 10.16 cm (4 inches) forward of the trailing edge and 20.3 (8 inches) past the trailing edge, respectively. The largest amplitude

response, greater than 10 dB compared to the other signals, was the 775 Hz mode. By placing one of the two microphones at the peak response location of the 775 Hz mode, moving the second microphone along the longitudinal axis and around the circumference of the nacelle, it was possible to determine the number of phase shifts of the standing mode on an oscilloscope. For the 775 Hz mode, there were three standing circumferential waves (n), and three standing longitudinal half waves (m), at 740 Hz $n=3$ and $m=2$ and at 790 Hz $n=3$ and $m=4$. No modes were uncovered in the radial direction. These frequencies agree with those predicted by the simple cylinder model with the 775 Hz mode identified as a strong candidate to be excited by a source located near the trailing edge of the strut. Although the $n=3, m=4$ mode is not shown in figure 6, it may be observed that this mode is probably in the same frequency range as the other $n=3$ standing modes.

Excitation mechanism.- Anticipating the source of the resonance excitations to be the vortex wake shed from the internal struts, several assumptions were made to estimate the frequency range of this source. The flow over the struts was probably turbulent because of several large recessions near the leading edge for support bolts. The chord Reynolds number of the struts is about 2×10^6 for the flight speeds of the test. Using the equations from chapter 9, reference 5, the frequency of the strut vortex shedding is in the range of 775 Hz. In reference 6, Archibald also found the most efficient acoustic source in exciting the mode was the rounded trailing edge.

Modification and effects.- Although the mode shape at 775 Hz in flight was not measured, Parker had observed both standing and traveling waves close in frequency in his experiment. It was, therefore, felt that this was a nacelle resonant mode excited by the aerodynamic flow over the support struts. Since the purpose of this test was to provide a method of removing the high amplitude frequency signal, it was felt that modifying the source would yield this result. By adding a tapered edge to the strut, the efficient curved edge would be eliminated; the boundary layer thickness at the trailing edge would change, moving the peak frequency of excitation and the location of this maximum input would move making it less likely to excite the mode. A tapered 3.175 cm (1-1/4") extension, made of balsa wood and covered with two layers of fiberglass, was bonded to the existing rounded-edge of the struts, figure 7, forming a relatively sharp trailing edge. The flight test was re-run slowly varying the aircraft speed from 61.7 m/s to 77.2 m/s (120 knots to 150 knots). The stepped sweep spectra, figure 8, shows no indication of the large resonant frequency at 775 Hz or any other frequency in the test range. The modification completely eliminated the problem.

CONCLUDING REMARKS

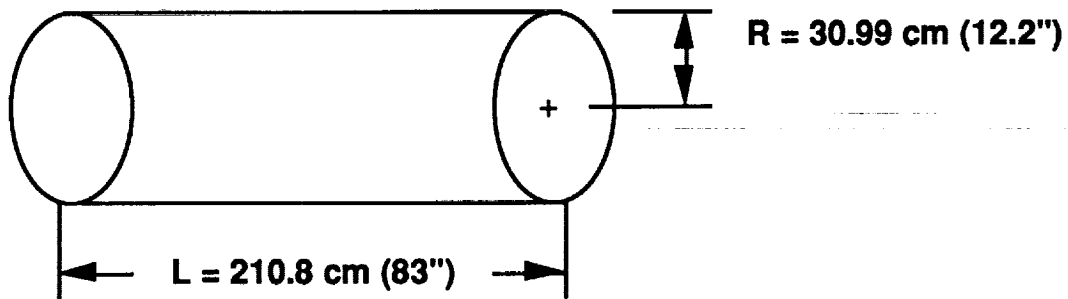
During an experimental investigation of laminar flow on the outside surface of a flow-through nacelle, simulating a high by-pass ratio engine nacelle, an intense acoustic signal was measured. The signal, which was neither anticipated nor desired, interfered with control conditions of the test. A brief investigation into the cause of the unwanted signal indicated that the source was a "Parker" mode, an interaction between an aerodynamics excitation and an acoustic mode of the nacelle. Based on

this evaluation, a modification to the trailing-edge of the support struts of the nacelle was made. This modification completely eliminated the problem and the flight test continued without any additional acoustic interference.

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TABLE 1.- Inputs and Results for Nacelle Acoustic Modes



$\frac{\bar{R}_{n0}}{\pi}$	$\frac{\bar{R}_{n1}}{\pi}$	n	Resonant Frequencies, Hz				
			s=0 m=1	s=1 m=1	s=0 m=2	s=1 m=2	s=3 m=3
0	1.2197	0	67	663	135	673	202
.5861	1.697	1	324	920	344	927	375
.9722	2.1346	2	530	1156	543	1162	563
1.3373	2.5513	3	726		736		750
1.6926	2.9547	4	917		925		937
2.0421	3.3486	5	1106		1112		1122

*From reference 3, Table 1

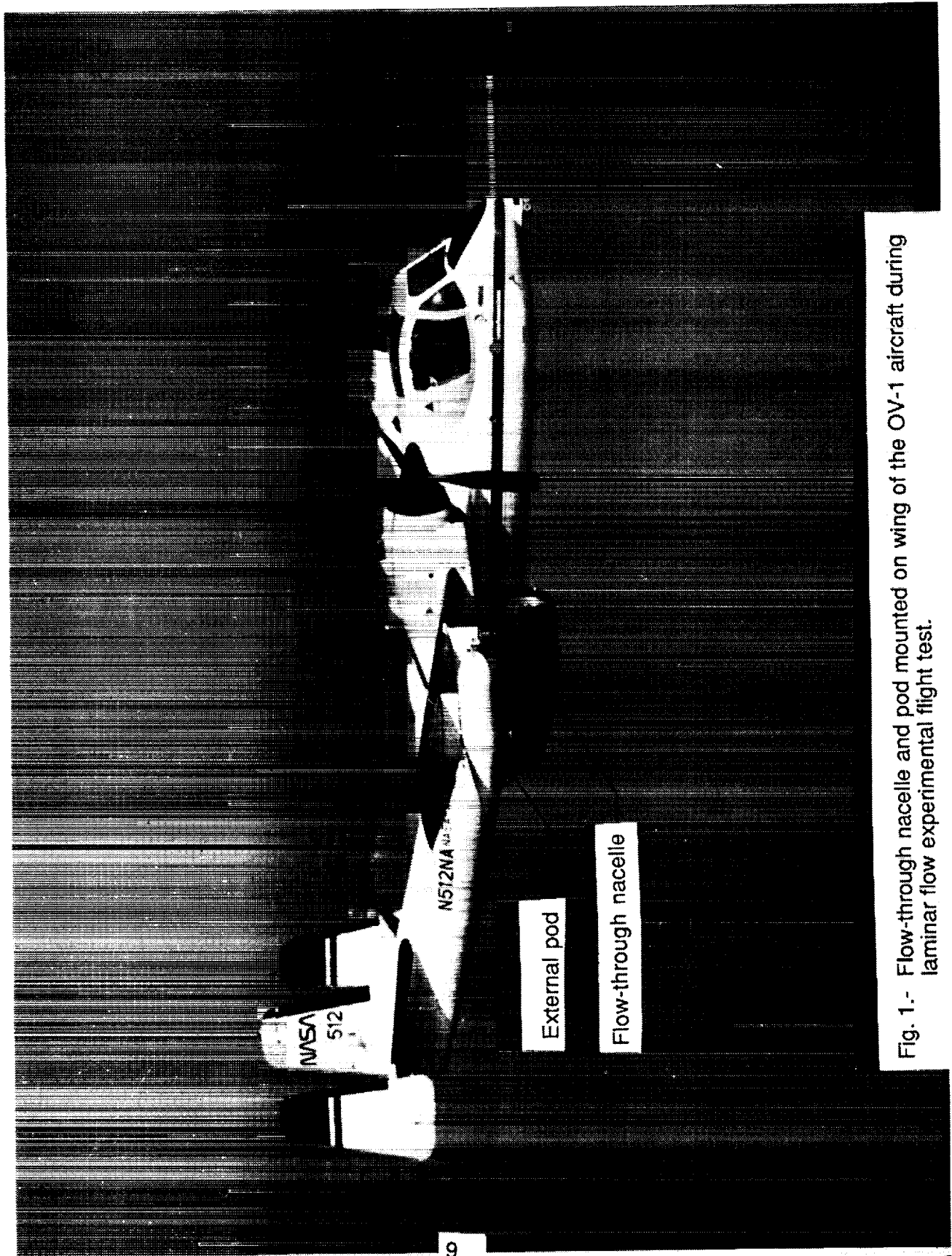
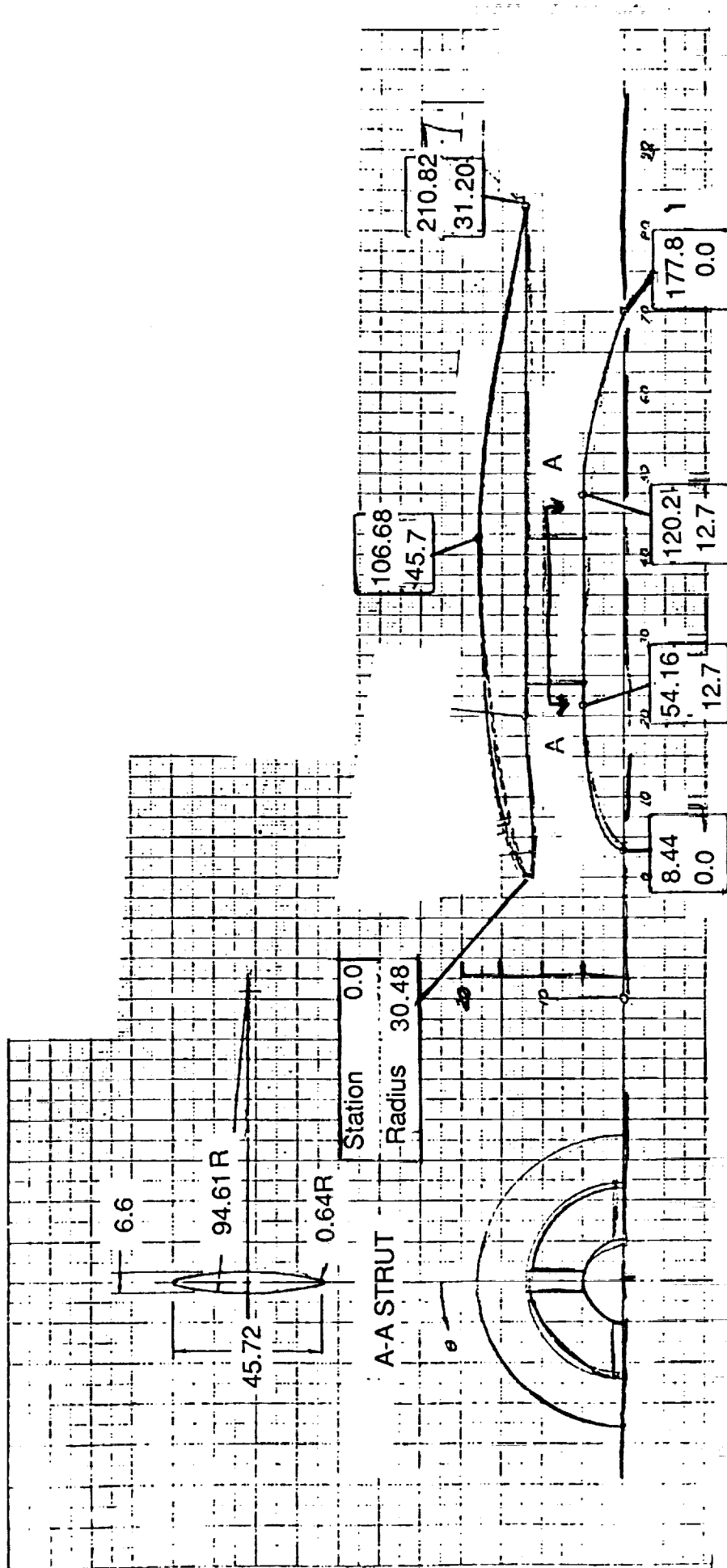


Fig. 1.- Flow-through nacelle and pod mounted on wing of the OV-10 aircraft during laminar flow experimental flight test.



All dimensions in centimeters

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Fig. 2.- Schematic view providing dimensions of the flow-through nacelle.

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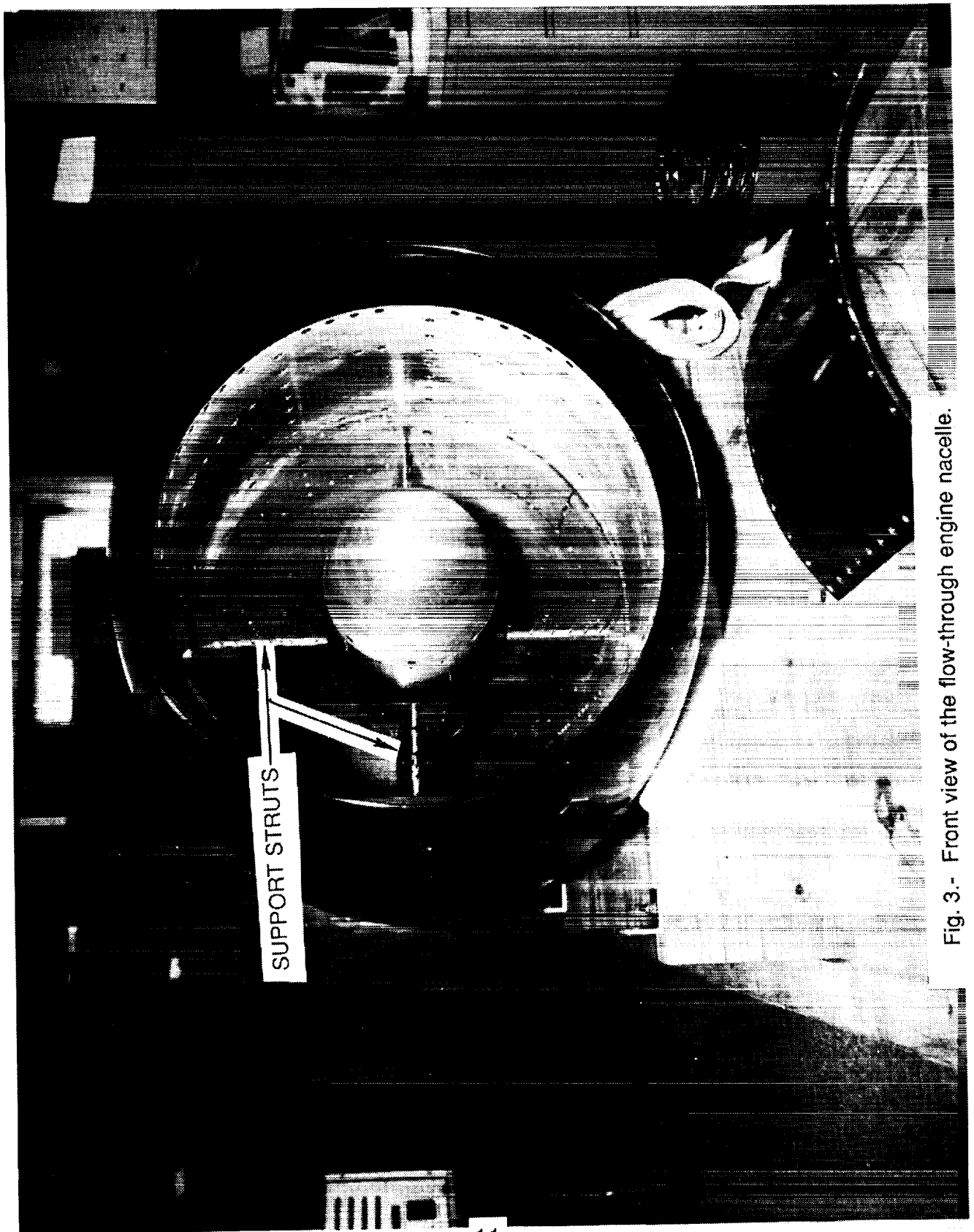


Fig. 3.- Front view of the flow-through engine nacelle.

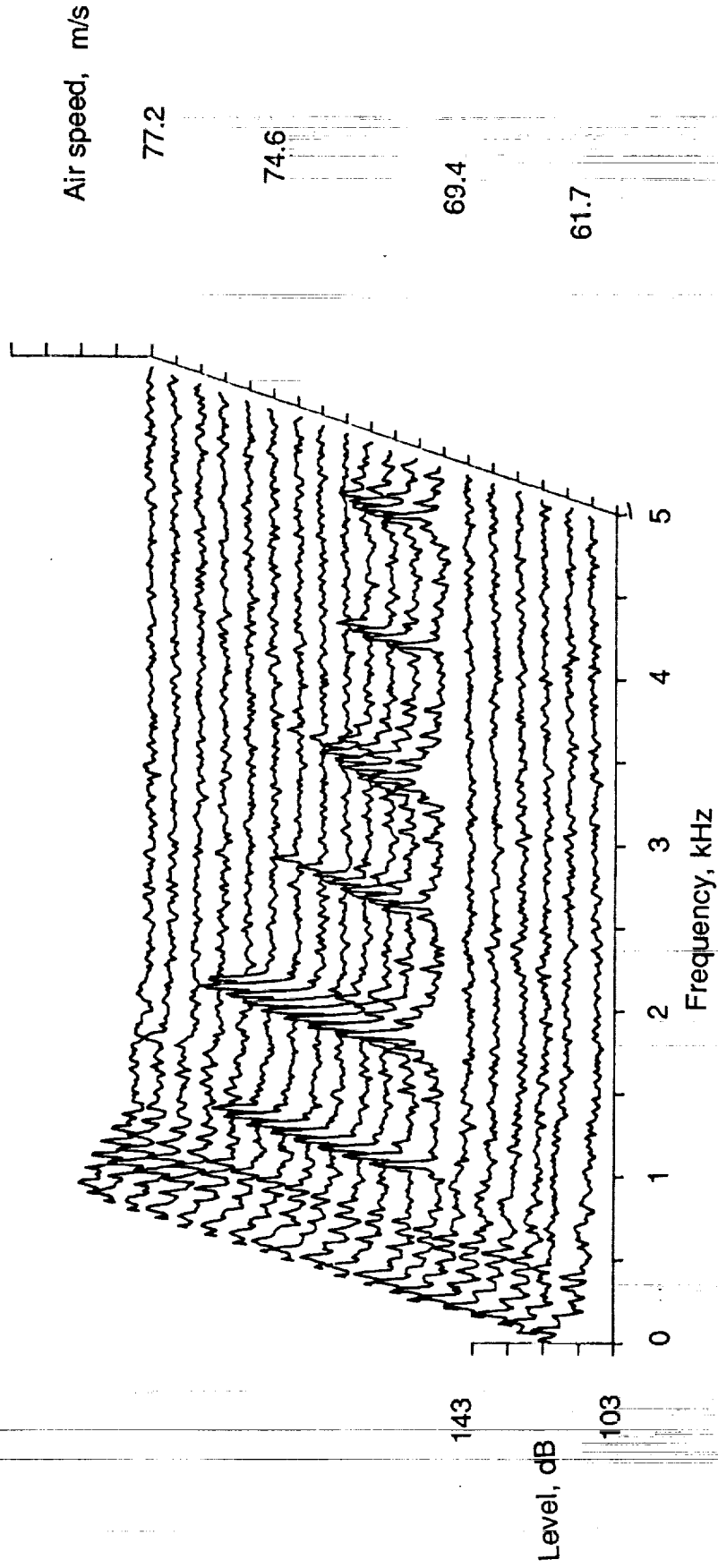


Fig. 4.- A stepped, sweep, time history of the pressure spectra before strut modification.

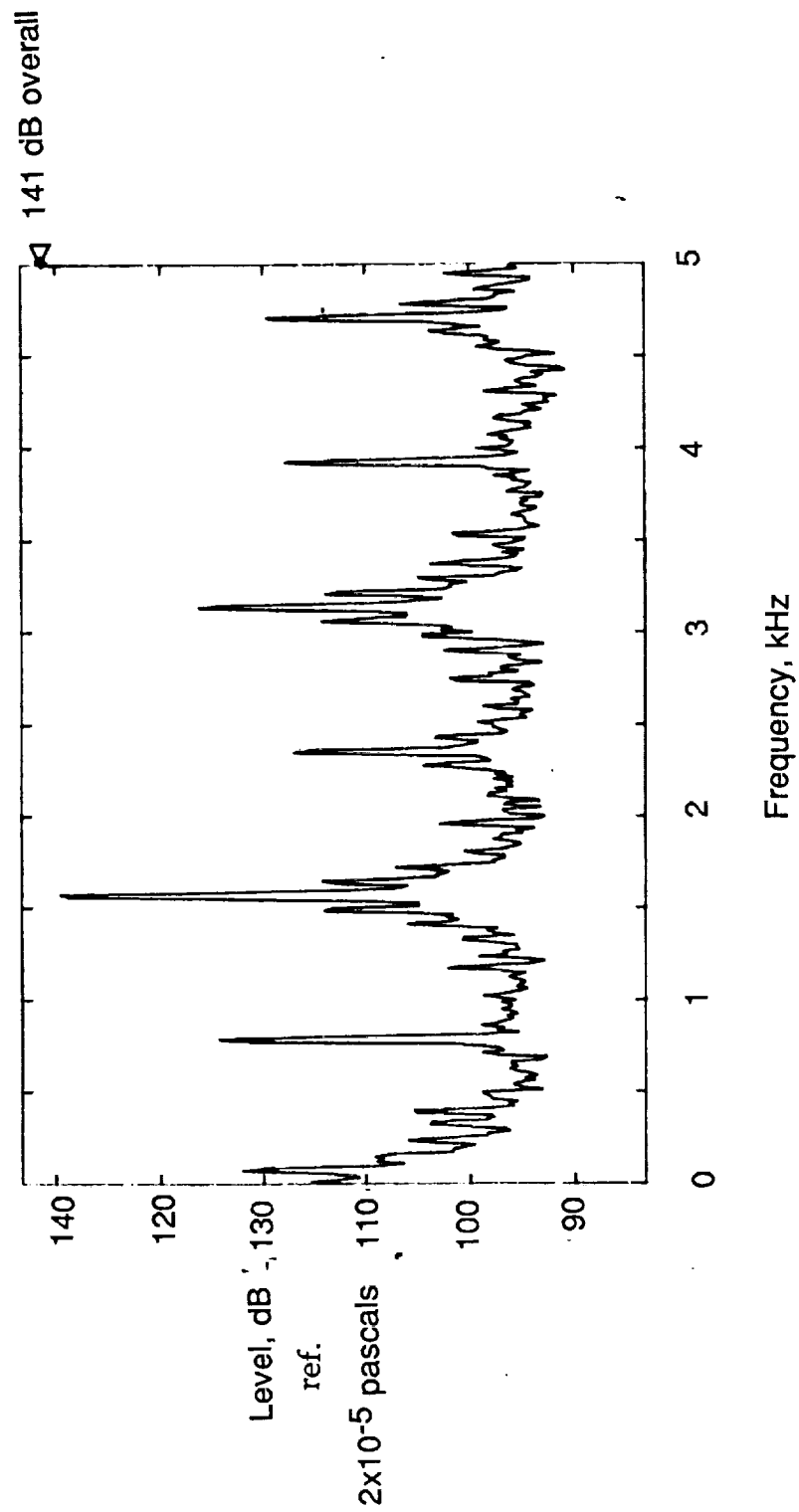


Fig. 5.- Sound pressure level plot from an internal transducer showing the frequency and levels of the fluctuating pressure at about 72.0 m/s (140 knots).

ACOUSTIC FREQUENCIES OF THE NACELLE

Resonant Frequency, Hz

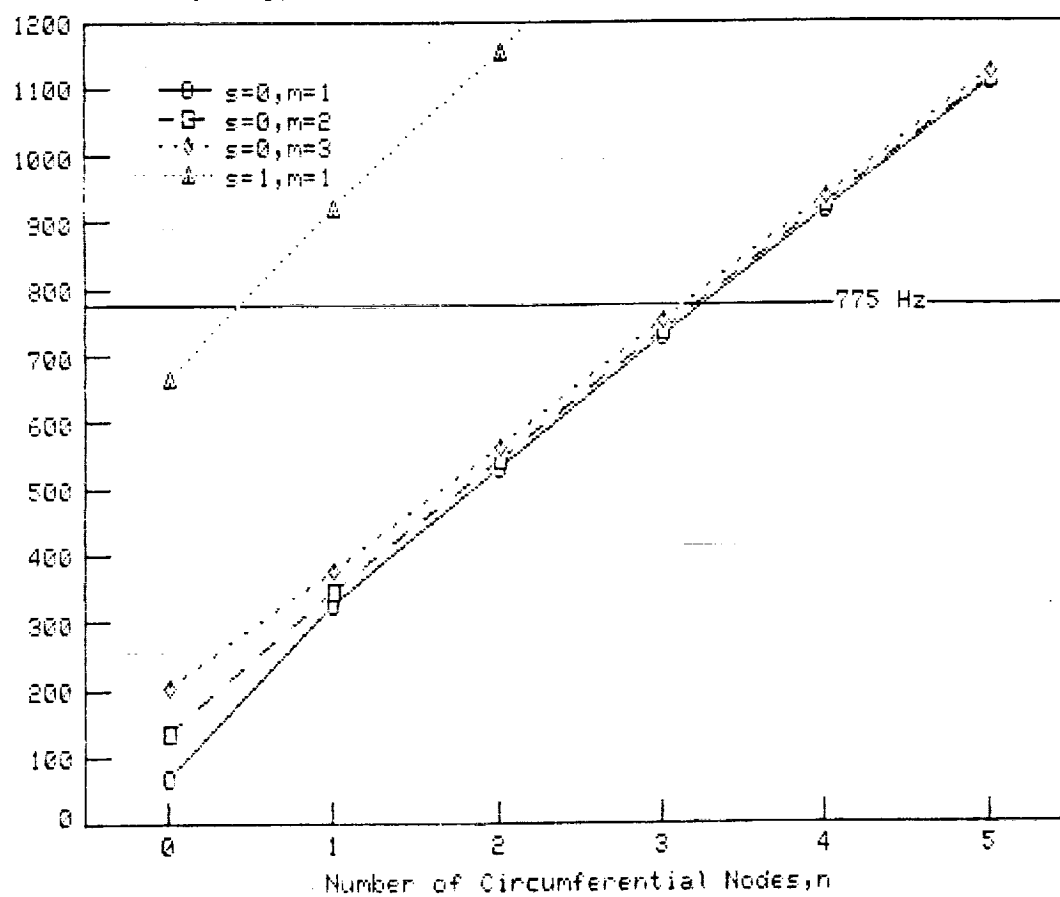


Fig. 6.- Plot of the coupled acoustic mode frequencies of the nacelle duct.



Fig. 7.- A view from the aft end of the nacelle showing the trailing edge modification.

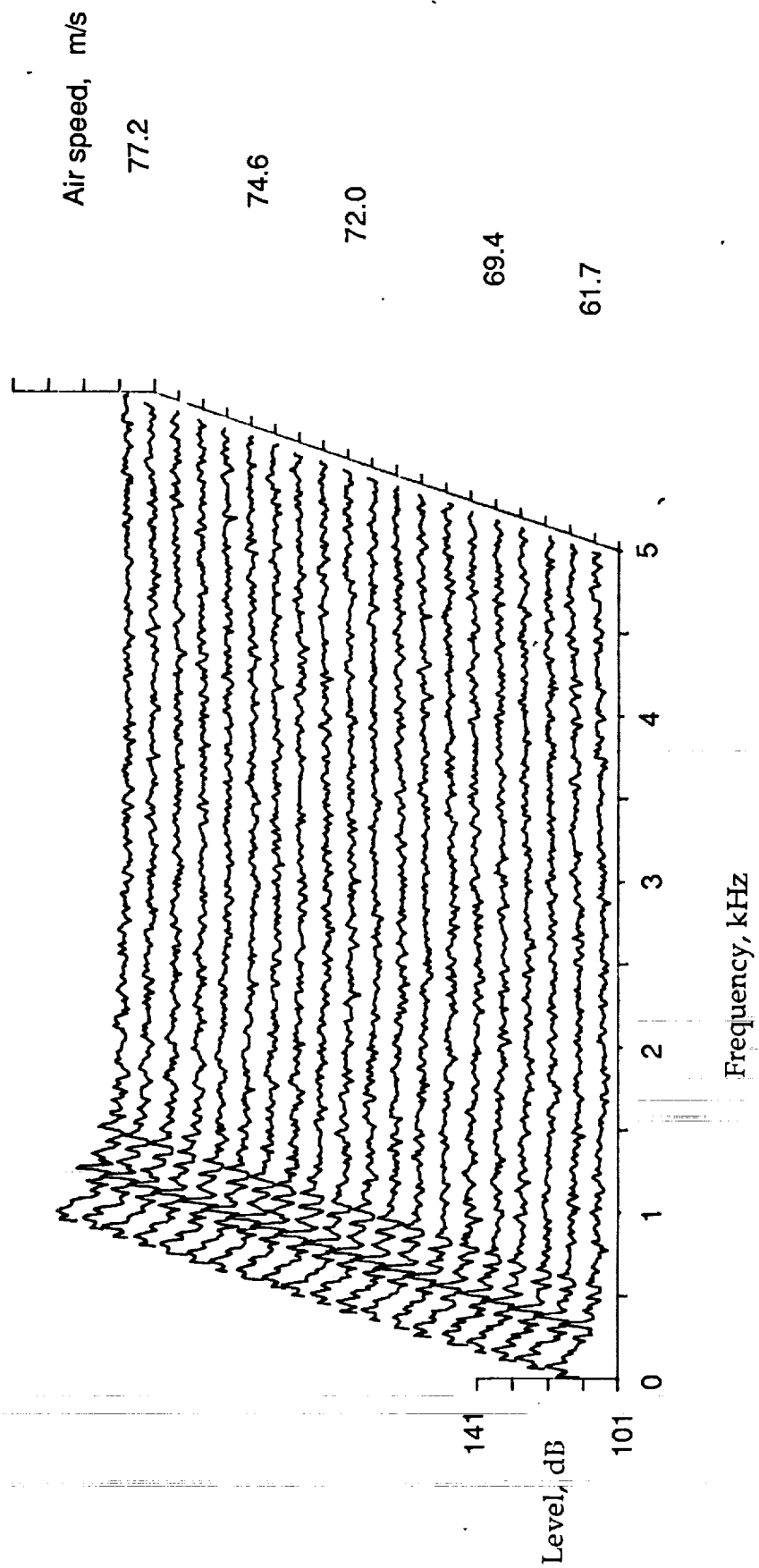


Fig. 8.- A stepped, sweep, spectral time-history after strut modification.



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